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RESEARCH MEMORANDUM

FLIGHT CALIBRATION OF ANGLE-OF-ATTACK AND SIDESLIP DETECTORS
ON THE FUSELAGE OF A 35° SWEEP-WING FIGHTER AIRPLANE

By Norman M. McFadden, George A. Rathert, Jr.,
and Richard S. Bray

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FLIGHT CALIBRATION OF ANGLE-OF-ATTACK AND SIDESLIP DETECTORS

ON THE FUSELAGE OF A 35° SWEEP-WING FIGHTER AIRPLANE

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SUMMARY

The position errors of angle-of-attack and sideslip detectors located on the fuselage of a 35° swept-wing fighter airplane were measured in flight over a Mach number range of 0.50 to 0.92 and at lift coefficients up to the buffet boundary. The variation of indicated angle of attack with true angle of attack was linear at each constant Mach number and altitude over the entire test range but both the slope and zero intercept varied with Mach number. It is shown that the angle of attack can be computed within $\pm 0.2^\circ$ on 90 percent of the data points using a linear equation in terms of indicated angle of attack and the ratio of impact pressure to static pressure. The variation of indicated angle of attack with sideslip angle was small and linear.

The indicated sideslip varied linearly with true sideslip and was insensitive to changes of Mach number, with the true sideslip being 63 percent of the indicated value.

INTRODUCTION

In many automatic fire-control and guidance systems for aircraft and guided missiles it is necessary to use angle-of-attack and sideslip input signals. Two general methods have been considered for obtaining these signals, continuously computing the true angle of attack from the known or measured dynamic pressure, gross weight, normal acceleration, and lift-curve slope, or measuring the angle of attack directly. Accuracies of the order of $\pm 0.2^\circ$ are usually required.

Turning our attention to the second method, in order to get an accurate measured indication of the true angle of attack it is necessary either to put a sensing device at a considerable distance in front of the

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aircraft to avoid the interference effects of the wings and fuselage, or to make corrections for these position errors if the sensing device is relatively near the aircraft. Nose or wing-tip booms of the order of 6 to 8 feet in length would be necessary to put the sensing device in an area relatively free of interference effects on airplanes the size of the present test vehicle. Besides introducing errors from bending of the booms, such long booms would be objectionable on operational aircraft. Therefore it seemed desirable to investigate a location on the fuselage nose where, although the true angle of attack could not be measured directly, the local angle of attack might be measured and corrected easily to give a signal proportional to the true angle of attack.

This report presents a flight-test calibration of the position error on the fuselage of a North American F-86A-5 airplane at a specific location suggested by the Aviation Ordnance Department of the Naval Ordnance Test Station at Inyokern, California. Although the numerical results obtained apply to a particular location on a particular airplane, it would be expected that similar results would be obtained on any body which is a reasonable approximation to a body of revolution, provided sufficient attention is given to avoiding local interference in selecting the location of the sensing device. The general form of the correcting equations should be the same with the constants changed to fit the particular aircraft and location.

NOTATION

- b empirical zero intercept of expression relating true and indicated angles of attack
- C_N normal-force coefficient $\left(\frac{\text{normal force}}{qS} \right)$
- C_{N_α} normal-force-curve slope $\left(\frac{\partial C_N}{\partial \alpha} \right)$
- g acceleration due to gravity, 32.2 feet per second per second
- H total pressure, pounds per square foot
- m empirical slope of expression relating true and indicated angles of attack
- M Mach number
- p static pressure, pounds per square foot
- q dynamic pressure $\left(\frac{1}{2} \rho V^2 \right)$, pounds per square foot

q_c impact pressure (H-p), pounds per square foot
 S wing area, square feet
 V true airspeed, feet per second
 α angle of attack, degrees
 β sideslip angle, degrees
 ρ density, slugs per cubic foot

Subscripts

I indicated
 T true
 C computed

EQUIPMENT AND TESTS

Basic Airplane and Instruments

The test airplane (fig. 1) was a standard North American F-86A-5. As a means of determining true angle of attack a nose boom, shown in figures 2 and 3, with five free-floating vanes located 20, 40, 60, 80, and 100 inches forward of the nose of the airplane was used. Figure 3 also shows the one-chord-length wing-tip booms on which the vanes used to determine true sideslip were mounted.

To measure the local angle-of-attack and sideslip angles on the fuselage, Specialties, Inc., Type J, Airstream Direction Detectors were located on each side of the fuselage and on the lower center line as shown in figures 4 and 5. These detectors are small cylindrical probes with two lengthwise slots spaced 60° apart which provide differential pressure to rotate the probe to seek the null or zero differential position which is recorded by a potentiometer. The slots, which are visible in figure 5(b), are $1\text{-}3/8$ inches long. The outer ends of the angle-of-attack slots were $3\text{-}13/32$ inches from the fuselage skin, and the outer ends of the sideslip slots were $2\text{-}1/8$ inches from the skin.

An 18-channel oscillograph, used to record all vane angles, was synchronized with standard NACA instruments recording impact pressure,

pressure altitude, and normal acceleration. A 16-millimeter motion-picture camera was used to photograph the bending deflection of the nose boom in flight.

The precision of the angle-of-attack vane measurements is estimated to be $\pm 0.1^\circ$; the precision of the sideslip-angle vane measurements, including unknown wing-tip boom deflections, is estimated to be $\pm 0.25^\circ$. The sensitivity of the Specialties Airstream Direction Detectors is specified as $\pm 0.11^\circ$ at values of dynamic pressure greater than 125 pounds per square foot, the minimum during the present tests.

Data Corrections and Tests

é? The angles of attack indicated by the five vanes on the nose boom were corrected for floating angle (due to asymmetry of the vanes), for upwash around the boom, and for the bending of the boom due to acceleration and air loads. The corrections for vane floating angle and boom upwash were obtained by calibrating the boom in the Ames 12-foot pressure wind tunnel in both the upright and inverted positions at Mach numbers from 0.50 to 0.96. Figure 6 shows, as a sample of the wind-tunnel data, that for the most forward vane the floating angle (one-half the difference between the angles indicated in the upright and inverted positions) is nearly independent of Mach number and angle of attack. The data at a Mach number of 0.96 have considerable scatter and are considered to be unreliable because of tunnel choking. Similar results were obtained for the other four vanes.

The upwash due to the boom was derived from figure 7 which presents the slope of the variation of indicated angle of attack of the forward vane with true angle of attack as a function of wind-tunnel dynamic pressure. The linearity of the data obtained at several Mach numbers indicates that there is no Mach number effect on the behavior of the vanes. The intercept at zero dynamic pressure, hence zero bending

deflection, $\frac{\partial \alpha_i}{\partial \alpha_T} = 1.058$, is the upwash correction. This corresponds

closely to the theoretical value of 1.062 from considerations of incompressible flow around a cylinder. The slope of the curve in figure 7 can be used to determine the boom deflection due to dynamic pressure; however, the flight results, which were affected by acceleration as well as by dynamic pressure, were corrected for the actual deflection of the boom determined by means of photographs taken during the test runs.

In addition to the above corrections, it was necessary to establish the amount of position error present ahead of the fuselage nose.

Figure 8 shows the indicated angle of attack of each vane as a function of the distance of the vane from the nose of the airplane for several normal-force coefficients at a Mach number of 0.81 and for several Mach numbers at normal-force coefficients of 0.10 to 0.35. Data for the vane 80 inches from the nose are omitted because of instrument malfunction. These curves are assumed to have reached their asymptotic values at the most forward vane which is 100 inches in front of the nose of the airplane. Thus this forward vane substantially indicates the true angle of attack. Consequently, this vane is used as indicating the true angle of attack of the airplane throughout the remainder of the report.

*ddi ~ 1.04
ddr ~ 1.04
detraining
upward at 100"*

The true angle of sideslip was obtained by averaging the readings of the vanes on the left and the right wing-tip booms. No corrections for unsymmetrical inflow at the wing tips due to sideslip were considered necessary.

The data from the flow-direction detectors mounted on the fuselage were corrected only for internal and external alignment with respect to the aircraft armament datum line.

Test flights were made at altitudes of 2,000, 10,000, 20,000, and 35,000 feet. The ranges of Mach number and normal-force coefficient covered in the investigation are shown in figure 9 with the buffet boundary of the test airplane included for reference. The sideslip was held to less than 1° for the angle-of-attack detector investigation.

RESULTS AND DISCUSSION

Angle of Attack

Figures 10, 11, and 12 present the variation of angle of attack indicated by the flow-direction detectors on the fuselage with true angle of attack at constant Mach number at each of the four test altitudes. The variations are linear over the entire range of angle of attack up to the buffet boundary. Over 80 percent of the data points are within $\pm 0.1^\circ$ of the faired straight lines, while over 96 percent of the points are within $\pm 0.25^\circ$. Because the small range of angle of attack covered at altitudes of 2,000 and 10,000 feet makes it impossible to define a slope accurately, the faired lines for these data (fig. 12) are averaged from the slopes and intercepts of the data from tests at altitudes of 20,000 and 35,000 feet (figs. 10 and 11).

Since the variations of true angle of attack with indicated angle of attack were shown by figures 10 through 12 to be linear at each Mach number, the true angle of attack can be expressed as

$$\alpha_T = m(\alpha_T) + b \quad (1)$$

The constants m and b are functions of Mach number and pressure altitude as shown in figure 13, which is a summary of the position-error calibration.

The slope, m , varies moderately from 0.63 at 0.50 Mach number to 0.58 at 0.88 Mach number, and increases abruptly to 0.66 at 0.92 Mach number as compared with a theoretical slope of 0.54 based on incompressible flow around a cylinder. The slopes are derived from the tests at high altitudes since the angle-of-attack range was limited at the low altitudes. The principal change in position error is the shift in the zero intercept b with Mach number from 0.5° at 0.60 Mach number to 1.3° at 0.92 Mach number.

It would be possible to use the slopes and zero intercepts given in figure 13 directly to correct indicated angles of attack for position error, although the computer required would be complicated by having to use nonlinear relationships. However, if the intercept b is plotted as a function of q_c/p , as in figure 14, a straight line through the origin is a reasonable representation of the experimental data. It would then be possible, by making the assumption that the slope m is independent of Mach number and altitude, to arrive at the relatively simple correction equation

$$\alpha_C = 0.615(\alpha_T) + 1.70(q_c/p) \quad (2)$$

For actual installations the constant 1.70 would be modified to correct for the position error of the airspeed installation used as the source of q_c and p . Reference 1 describes the F-86 airspeed system calibration.

To check the accuracy of equation (2), it was used to calculate the true angle of attack for the data from these tests. Figure 15 presents these results as a function of true angle of attack. The line of perfect correlation and lines of $\pm 0.2^\circ$ deviation are shown for reference. More than 90 percent of the points are within the desired $\pm 0.2^\circ$.

In order to show the data of figure 15 in more detail, the data have been plotted in figure 16 as the difference between calculated and true angles of attack as a function of true angle of attack. Separate curves are presented for the three ranges of test altitude.

In addition to the effects of Mach number and normal-force coefficient, the effect of sideslip on the angle-of-attack position error was investigated. Figure 17 shows the variation (for the detector on the right side of the fuselage) of indicated angle of attack with

sideslip at several constant values of true angle of attack and at several Mach numbers. The changes in position error with sideslip are small, about 0.1° change in angle of attack for each degree of sideslip. The significance of this change, of course, depends upon the actual amount of sideslip obtained in flight and will have to be evaluated in each case. Since the variation is linear through zero true sideslip, the use of the averaged signal from two detectors, one right and one left, would eliminate this effect.

Sideslip

The position error of the sideslip installation is presented in figure 18 for angles of attack of 1° to 5° . The variation of indicated sideslip angle with true sideslip angle is apparently linear and unaffected by Mach number. Within the estimated accuracy of measurement of true sideslip angle ($\pm 0.25^\circ$) the data shown in figure 18 can be adequately represented by the equation

$$\beta_T = 0.63 \beta_I \quad (3)$$

CONCLUSIONS

The measurement of the position errors of angle of attack and sideslip detectors ~~on the fuselage~~ of a 35° swept-wing airplane have indicated:

1. Over the test range for the angle-of-attack calibration, which extended up to the buffet boundary of the test airplane at Mach numbers from 0.50 to 0.92, the indicated angle of attack varies linearly with true angle of attack. Over 80 percent of the test points were within $\pm 0.1^\circ$ of faired straight lines, while over 96 percent were within 0.25° .

2. The true angle of attack was obtained to $\pm 0.2^\circ$ on 90 percent of the data points by correcting the indicated data with the equation

$$\alpha_T = 0.615 \alpha_I + 1.70 \frac{q_c}{p}$$

3. For an angle-of-attack detector mounted on one side of the fuselage there was a small linear variation of indicated angle of attack with sideslip amounting to 0.1° of angle of attack for each degree of sideslip angle. This effect could be eliminated by using the averaged signal of two detectors, one on each side.

4. Over the range of angles of attack corresponding to level flight, approximately 1° to 5° , and between Mach numbers of 0.70 and 0.91, the variation of indicated sideslip with true sideslip was linear and was independent of Mach number. The true sideslip angle was 63 percent of the indicated sideslip angle.

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REFERENCES

1. Thompson, Jim Rogers, Bray, Richard S., and Cooper, George E.: Flight Calibration of Four Airspeed Systems on a Swept-Wing Airplane at Mach Numbers up to 1.04 by the NACA Radar-Phototheodolite Method. NACA RM A50H24, 1950.

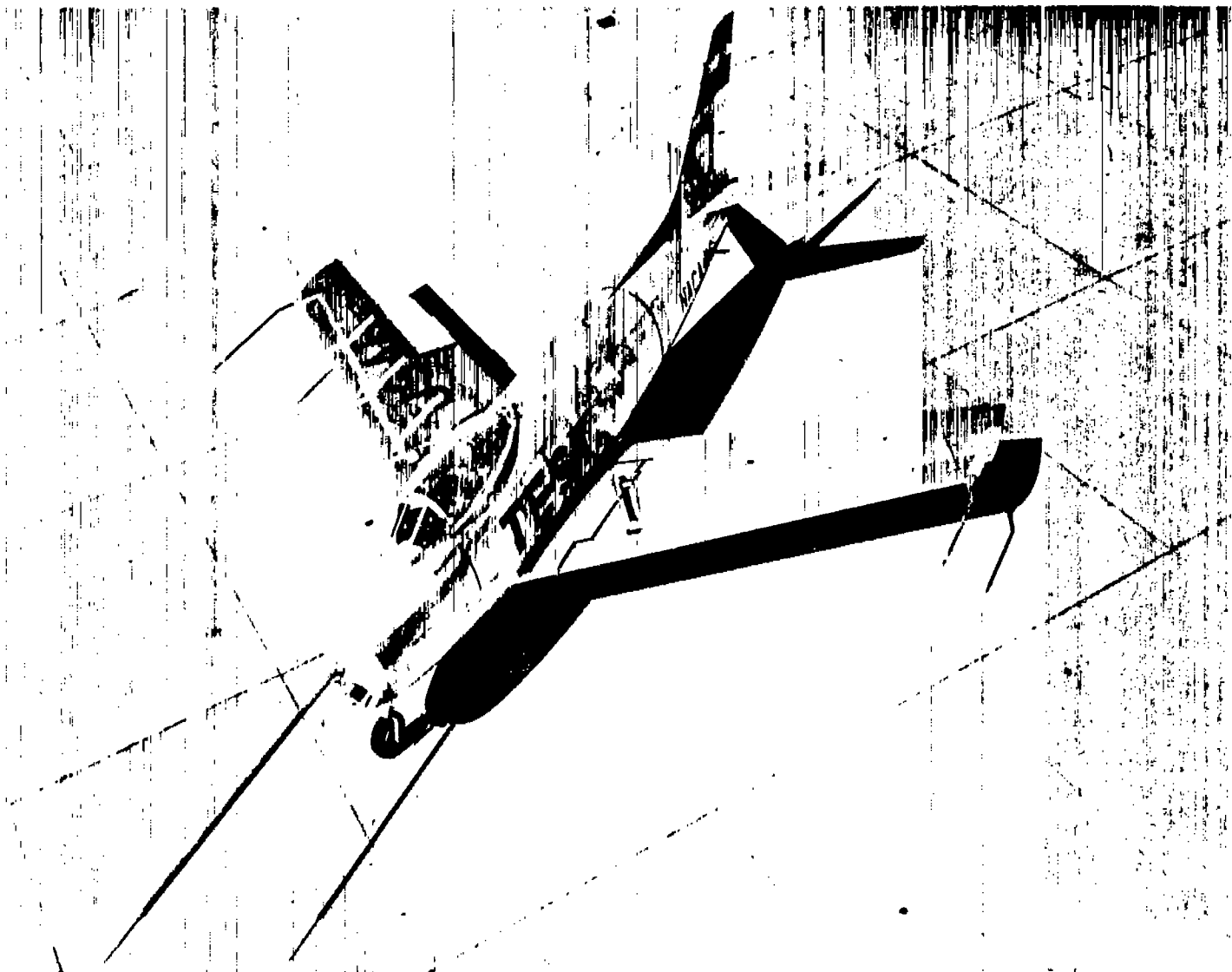


Figure 1.— The test airplane.

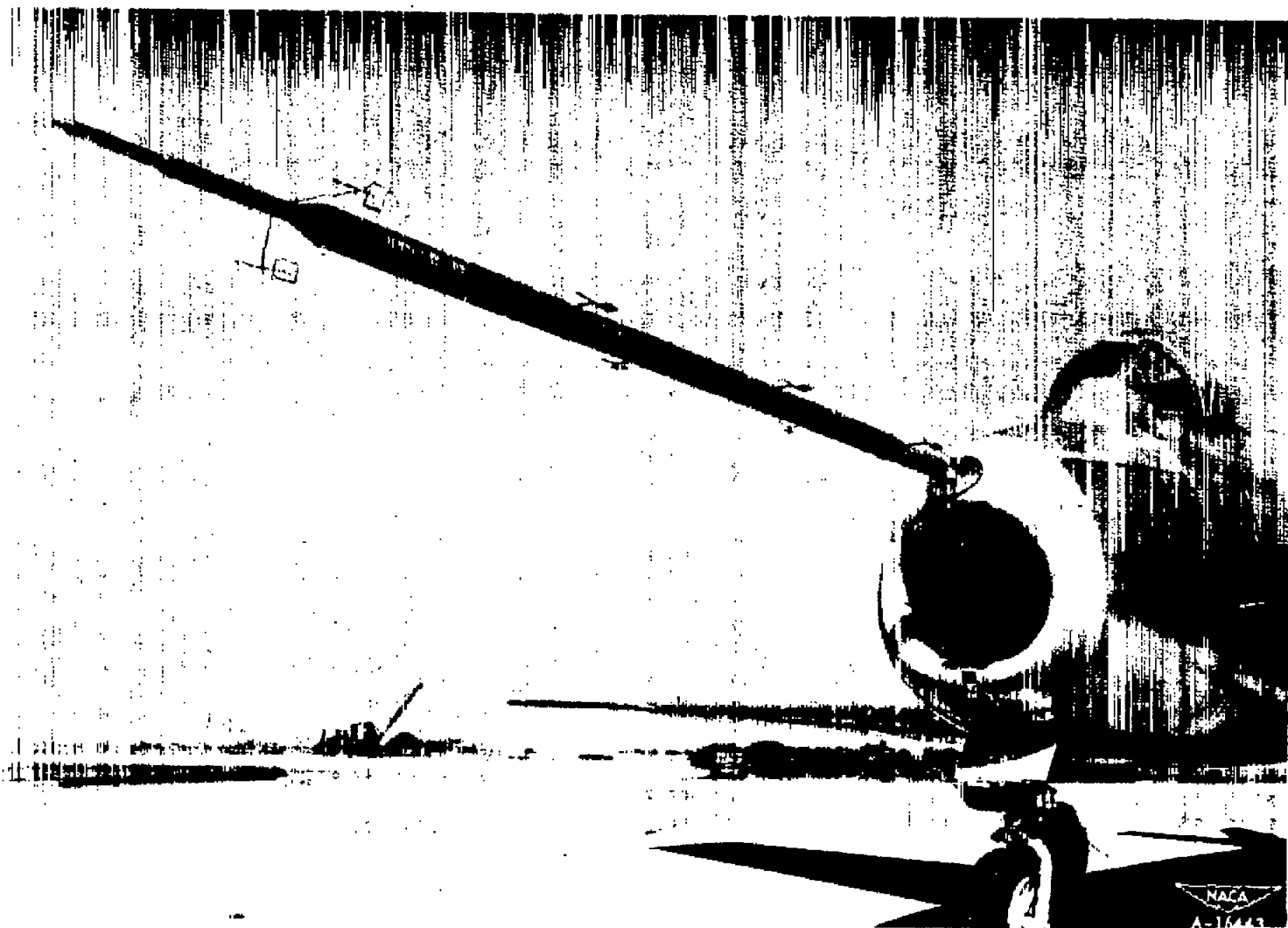


Figure 2.-- The nose-boom installation with angle-of-attack vanes.

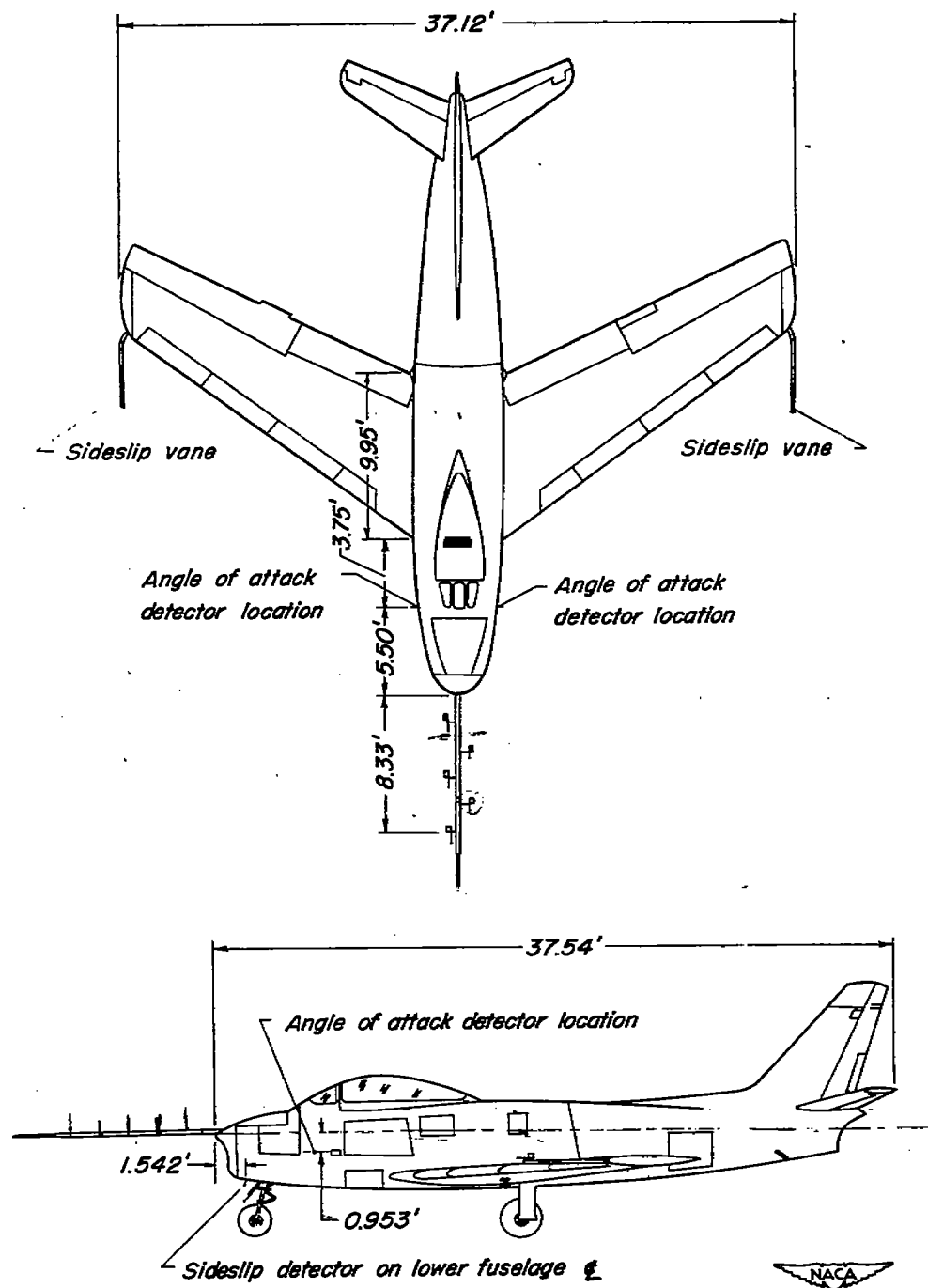


Figure 3.-Two-view drawing of the test airplane.

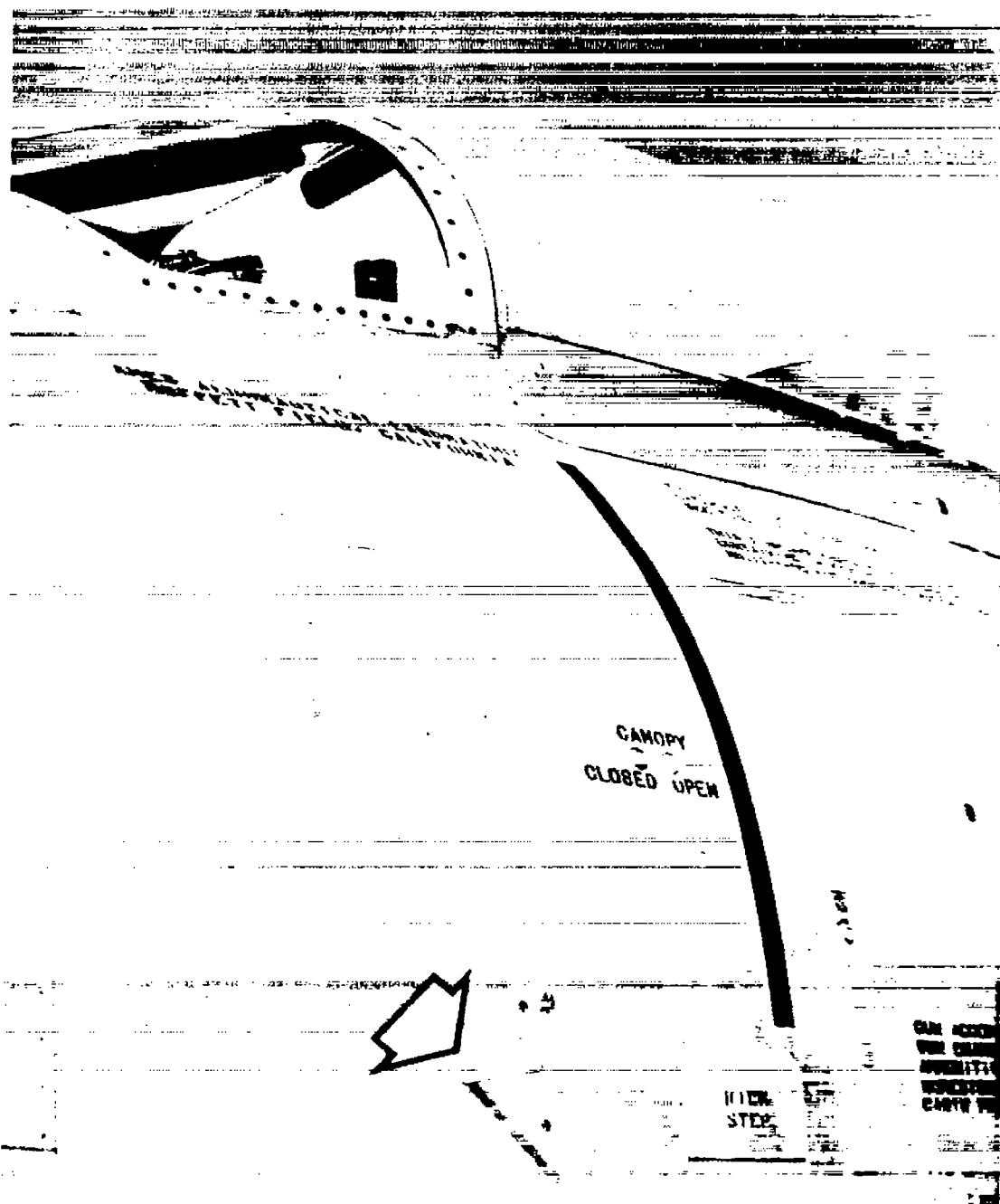
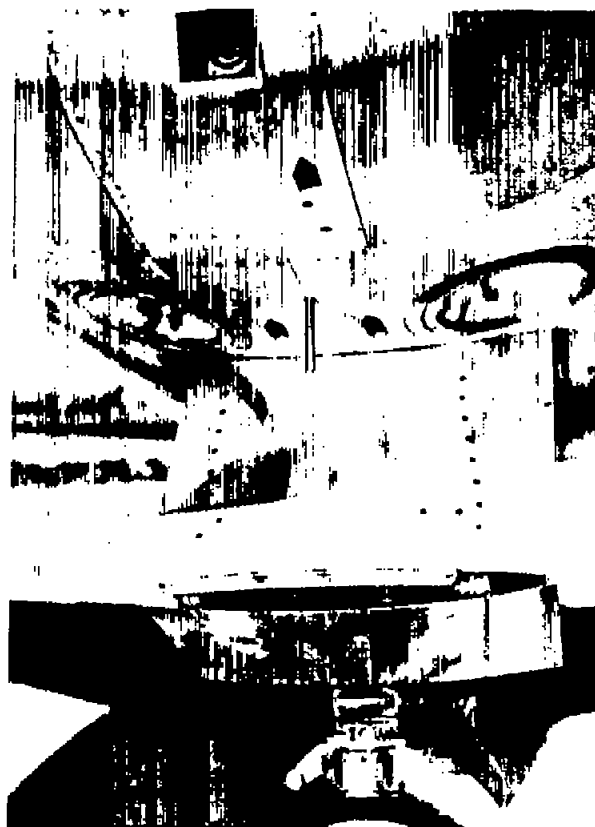
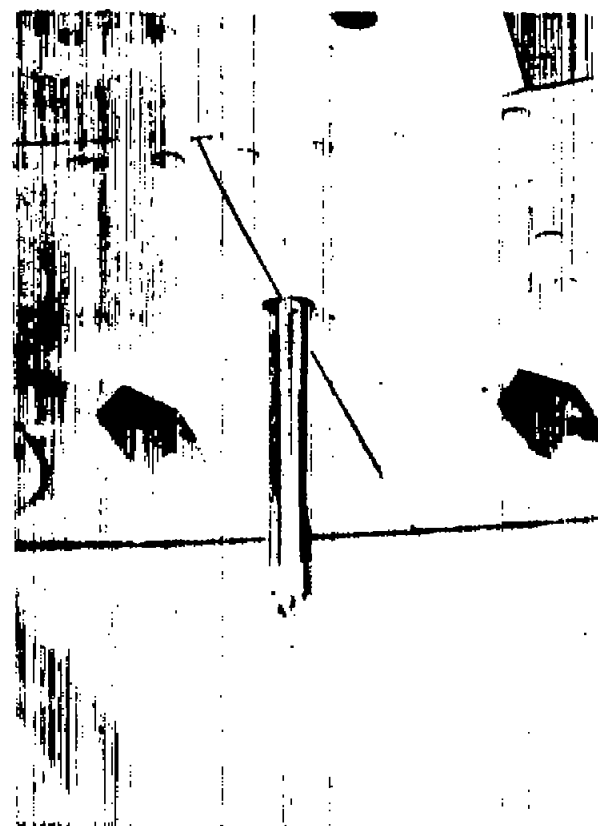


Figure 4.— The angle-of-attack detector on the left side of the fuselage.



NACA
A-16445

(a) Location of the detector.



NACA
A-16445.1

(b) Close-up of the detector.

Figure 5.- The angle-of-sideslip detector.

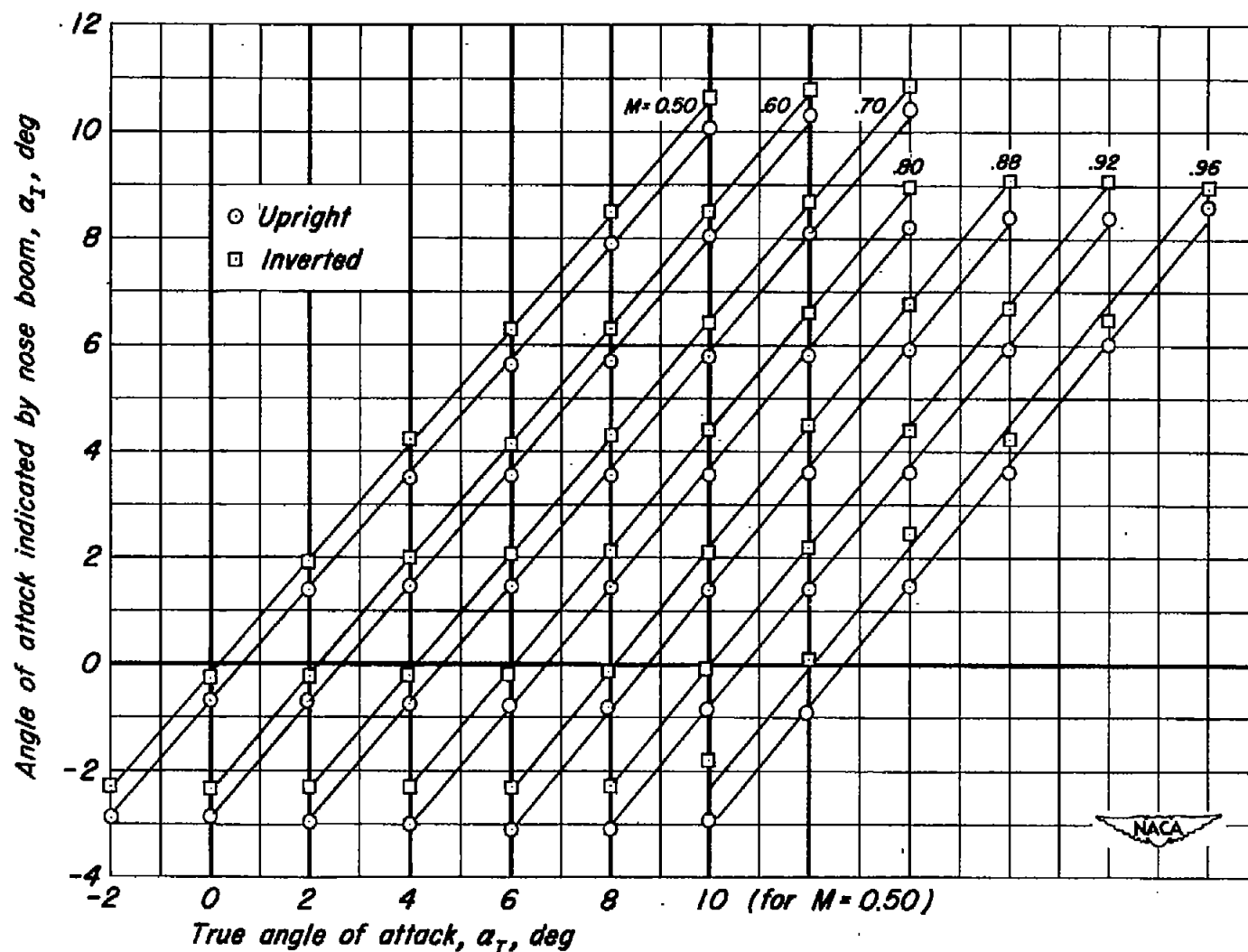


Figure 6.—The variation of the angle of attack indicated by the nose-boom vane with true angle of attack at various Mach numbers. Ames 12-foot pressure wind tunnel tests.

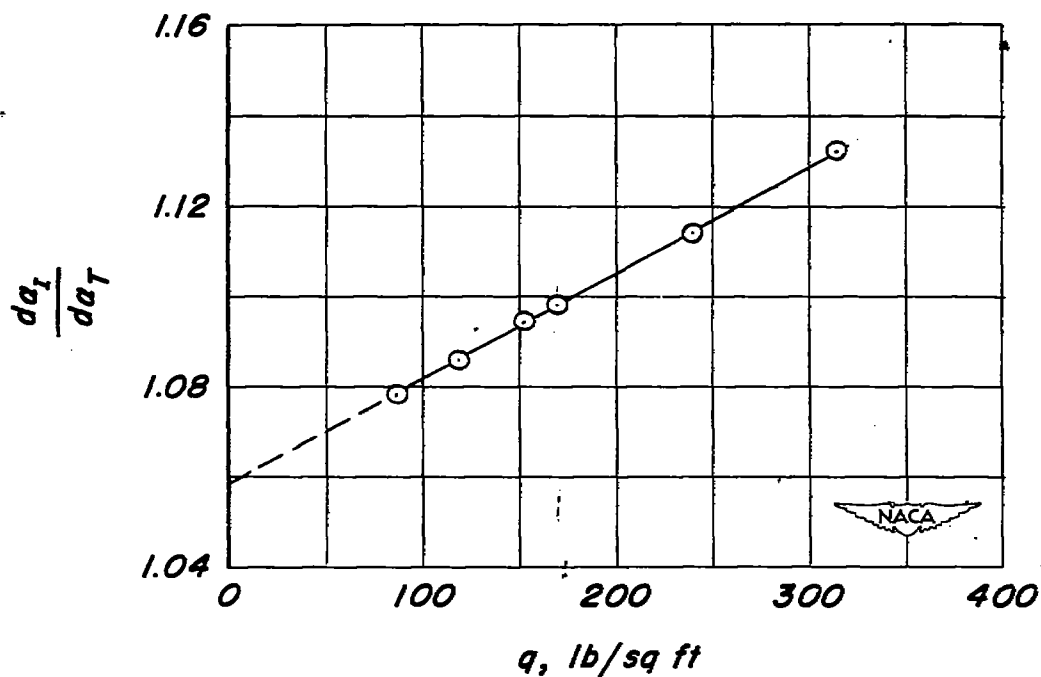


Figure 7.— The variation with dynamic pressure of the rate of change of the angle of attack indicated by the forward vane of the nose boom with the true angle of attack as measured in the Ames 12-foot pressure wind tunnel. (upwash due to boom only).

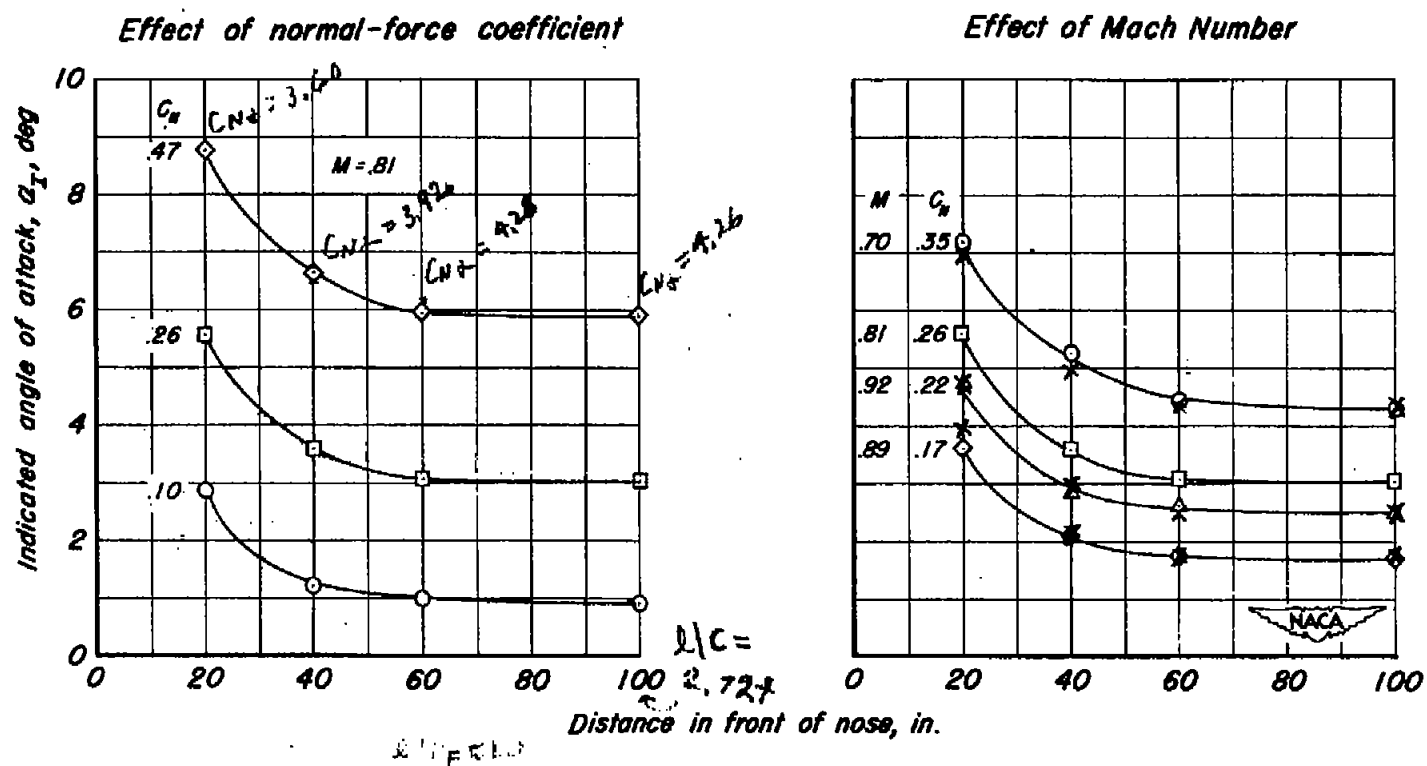


Figure 8.—Variation of indicated angle of attack with distance in front of nose of airplane.

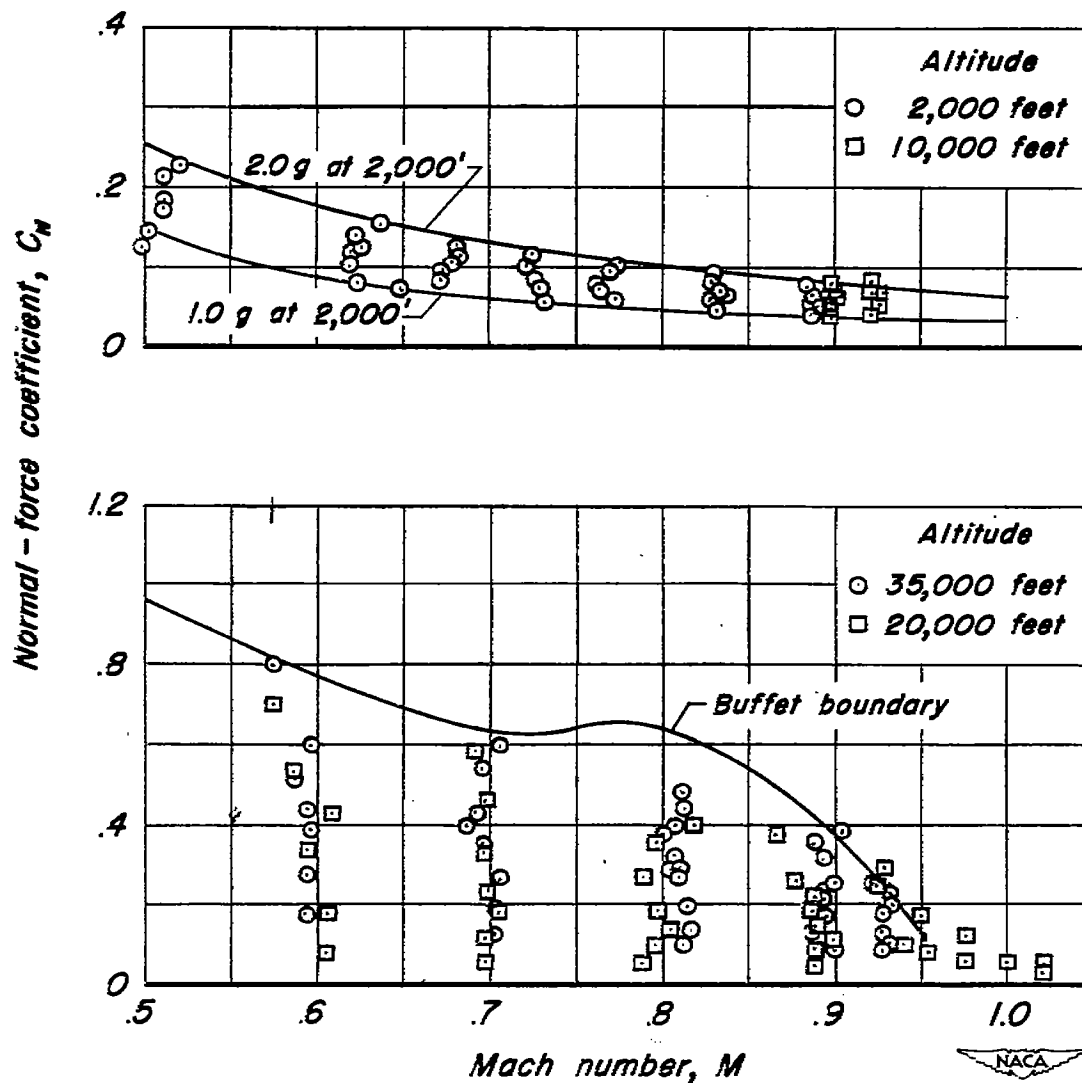


Figure 9.— The ranges of Mach number, normal-force coefficient, and altitude included in the test program.

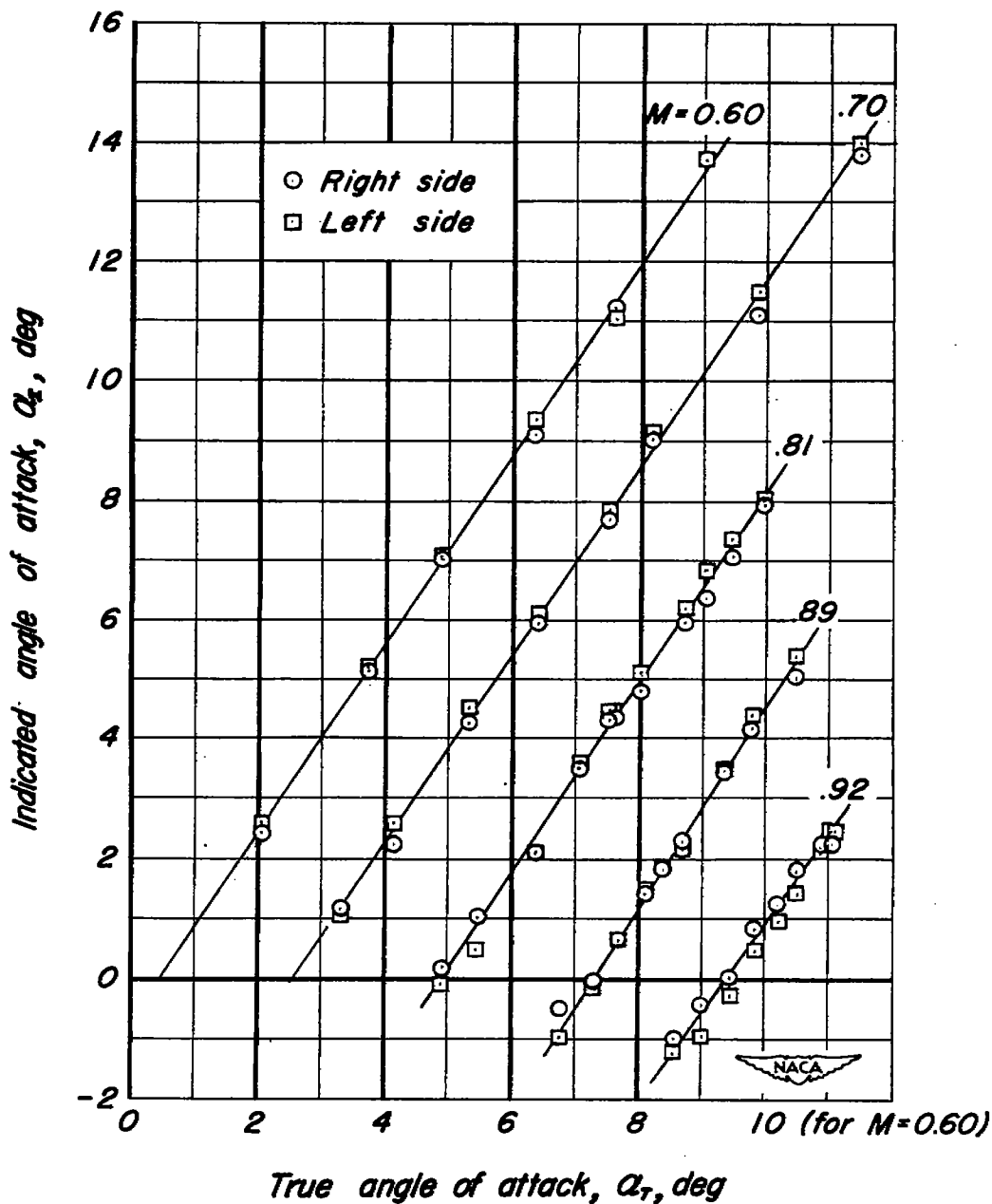


Figure 10.— The variation of indicated angle of attack with true angle of attack at several Mach numbers and 35,000 feet altitude.

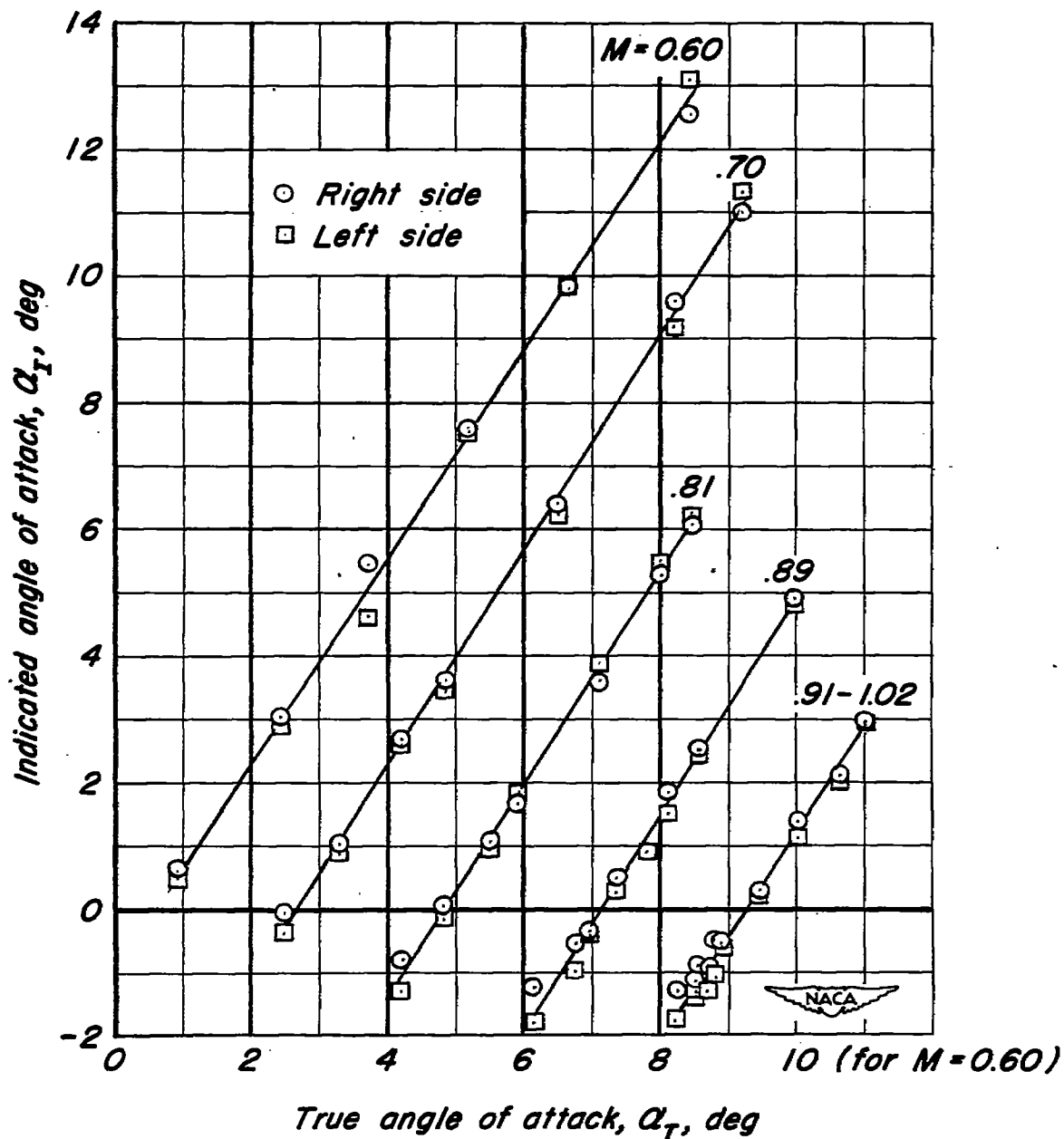


Figure 11.—The variation of indicated angle of attack with true angle of attack at several Mach numbers and 20,000 feet altitude.

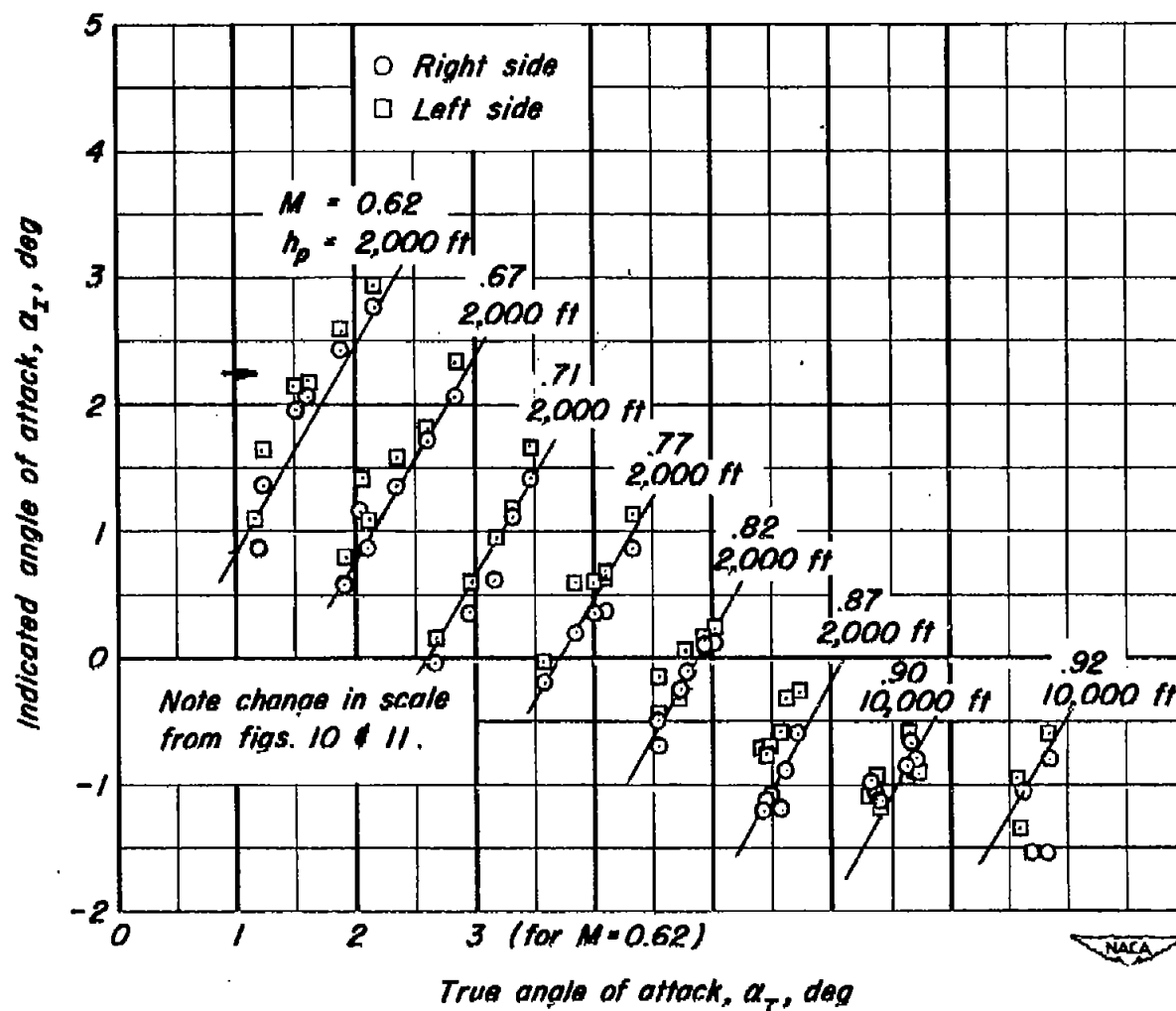


Figure 12.—The variation of indicated angle of attack with true angle of attack at several Mach numbers and altitudes of 2,000 and 10,000 feet.

Expression for a_{TRUE} :

$$a_T = m a_I + b$$

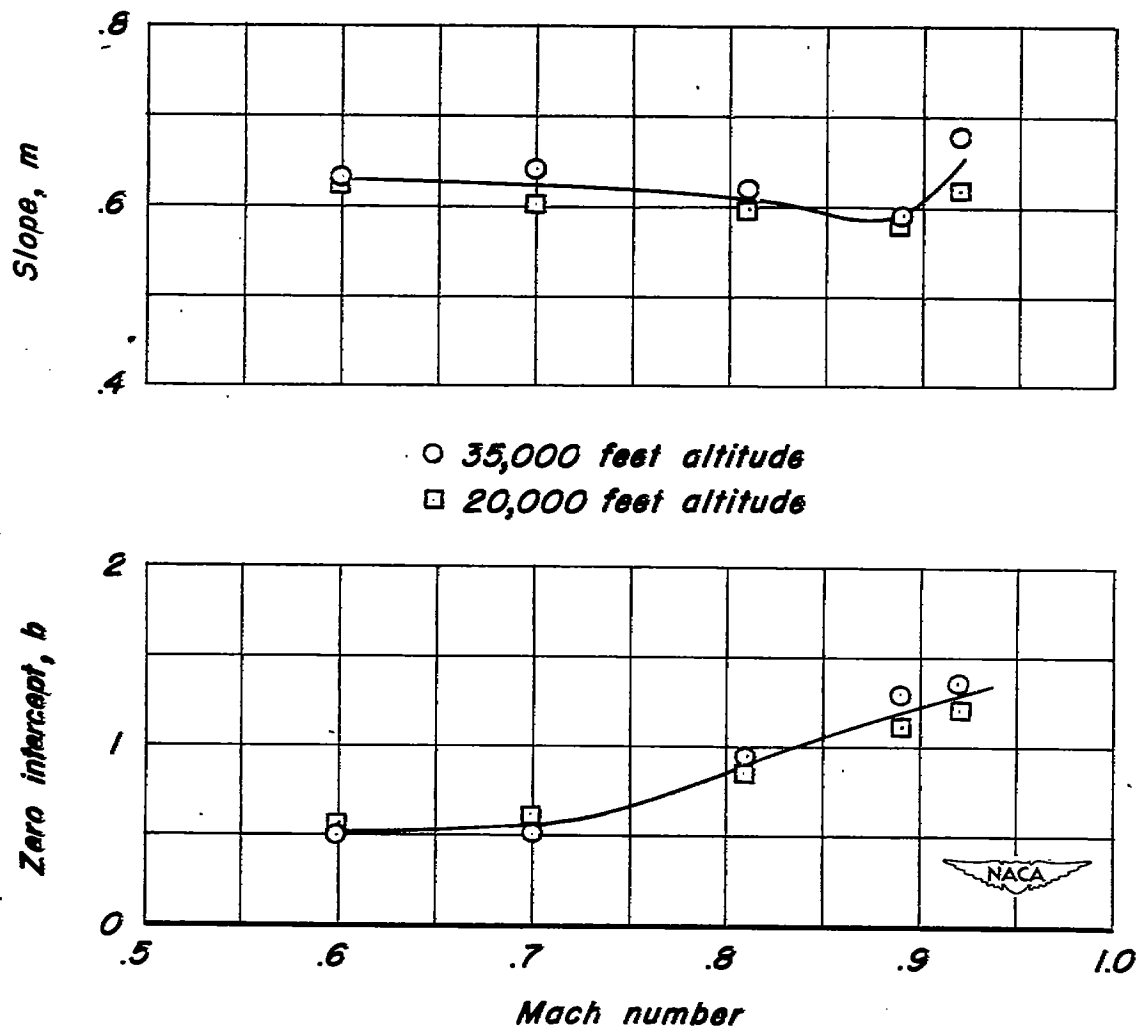


Figure 13.— The variation with Mach number of the slope and zero intercept of the expression relating the true and indicated angles of attack.

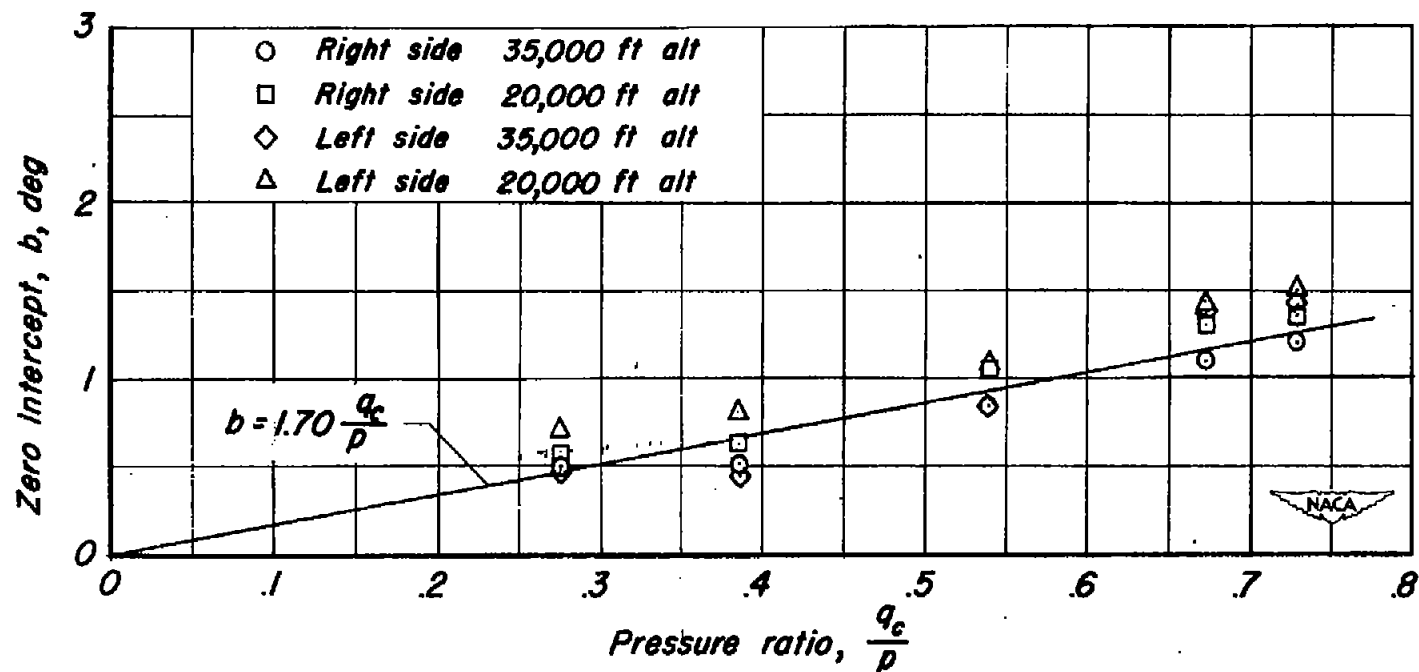


Figure 14.—The variation of the zero intercept b with the pressure ratio $\frac{q_c}{p}$.

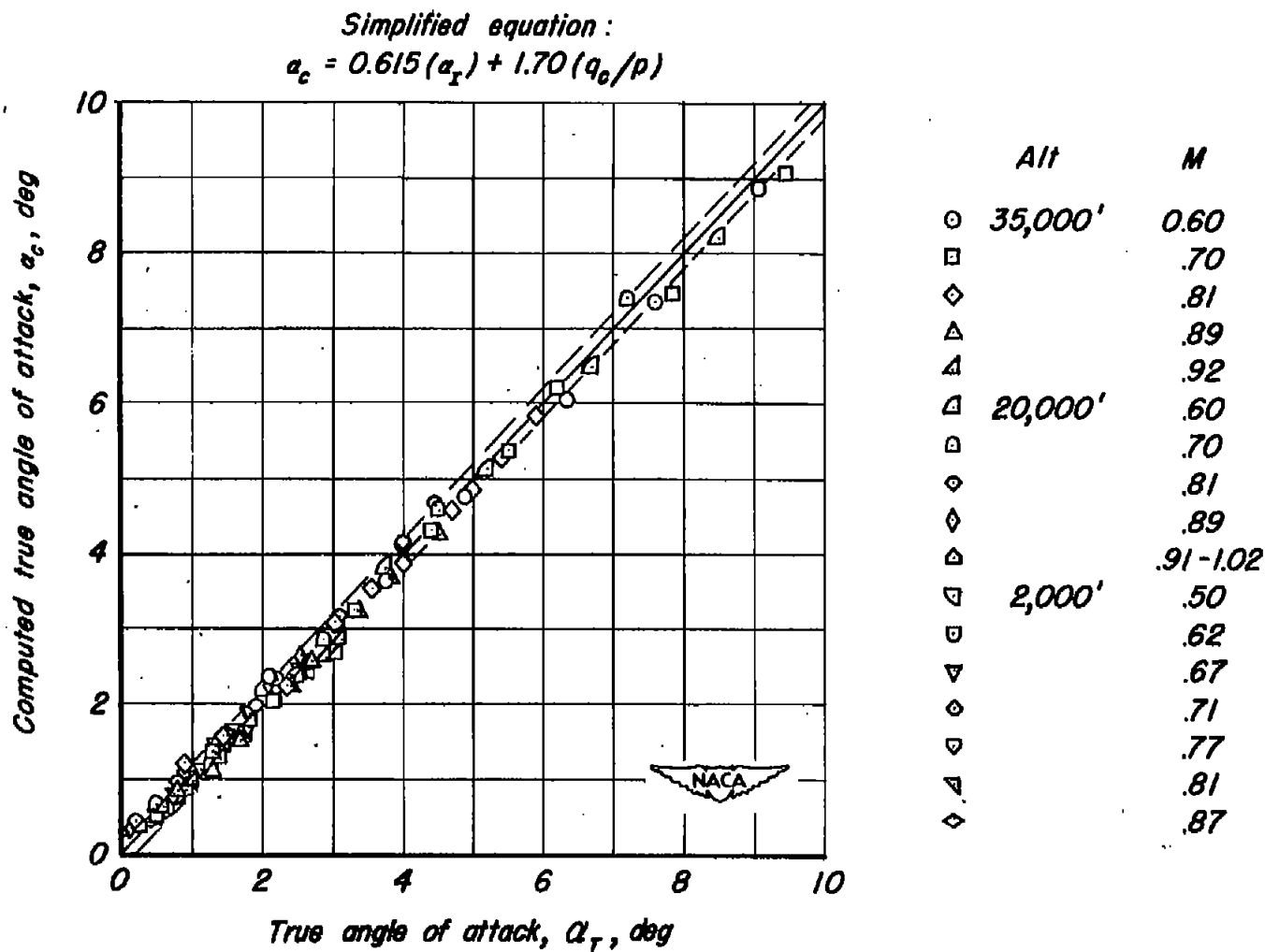


Figure 15.— The comparison of the true angle of attack with that computed using a simplified equation.

Altitude, 35,000 ft 20,000 ft 2,000 ft

	<i>M</i>		<i>M</i>
○	.60	△	.50
□	.70	□	.62
◇	.81	◇	.67
△	.89	◇	.71
△	.92	△	.77
		◇	.87

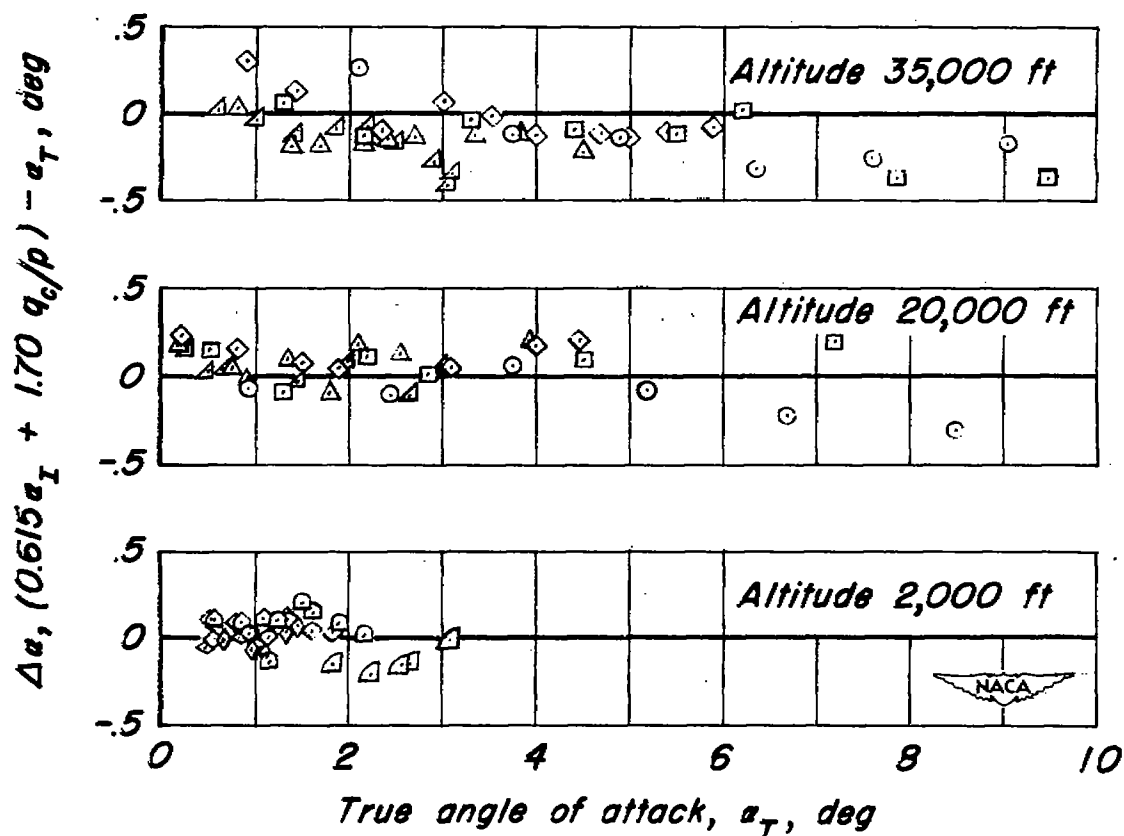


Figure 16.- The difference between the measured and computed values of angle of attack as a function of true angle of attack at several altitudes.

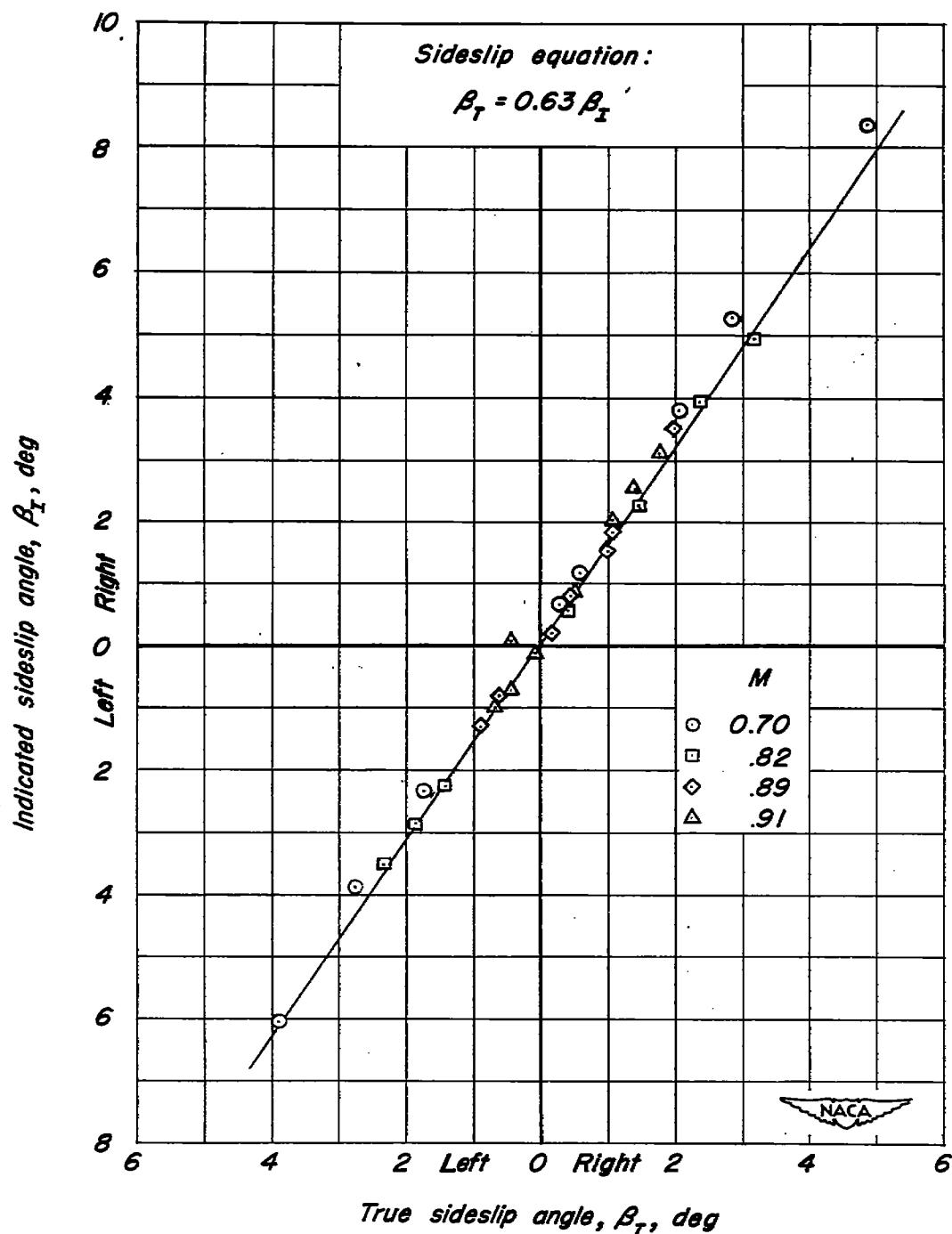


Figure 18.-The variation of indicated sideslip angle with true sideslip angle at several Mach numbers and 35,000 feet altitude.

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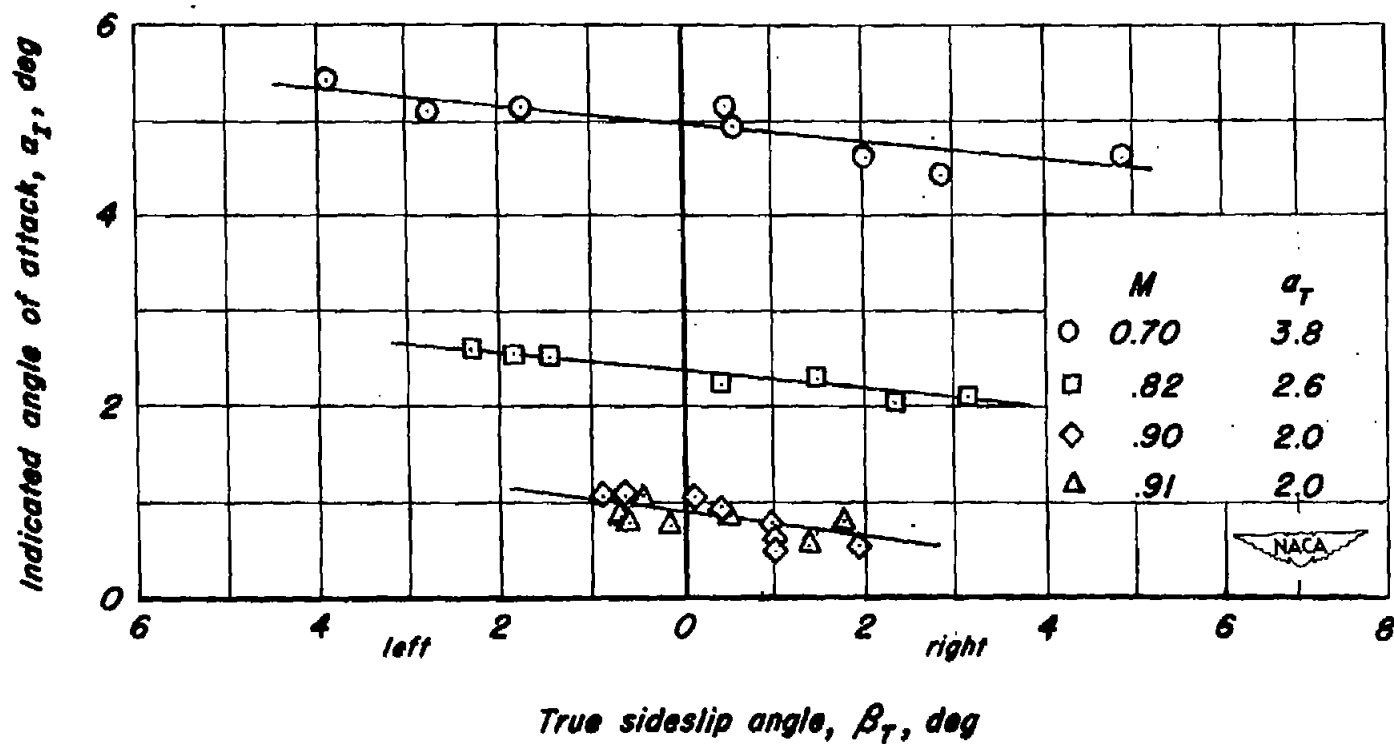


Figure 17.—The effect of sideslip on the angle of attack indicated by the right detector for several values of Mach number and at constant angles of attack.